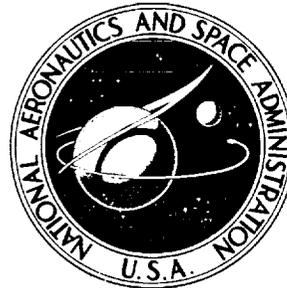


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APOLLO EXPERIENCE REPORT -  
AEROTHERMODYNAMICS EVALUATION

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16. Abstract The Apollo Program offered the first opportunity to obtain aerothermodynamic measurements at superorbital velocities on full-scale spacecraft. Four unmanned flight tests were conducted to qualify the Apollo command module heat shield. Aerothermodynamic measurements were made, and data are presented in this paper to illustrate the comparison of the flight data with the ground-test results and theoretical predictions.			
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# APOLLO EXPERIENCE REPORT

## AEROTHERMODYNAMICS EVALUATION

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### SUMMARY

Aerothermodynamic measurements were obtained on Apollo spacecraft during atmospheric entry at both orbital and superorbital velocities. These aerothermodynamic measurements confirmed the predicted Apollo environment that was used for the design of the thermal protection system (ablator) of the spacecraft. The Apollo Program offered the first opportunity to obtain measurements on full-scale spacecraft at superorbital velocities. This report contains the information gained of the Apollo spacecraft environment at superorbital and orbital velocities.

Data were obtained during four unmanned Apollo heat-shield-qualification flight tests. Measurements were made with pressure transducers, radiometers, and surface-mounted calorimeters. Selected flight measurements are compared with wind-tunnel data and theoretical predictions to illustrate the effectiveness of the existing prediction techniques. The aerothermodynamic analysis is indicative that wind-tunnel data can be used to describe the pressure environment of the Apollo spacecraft during entry, that laminar cold-wall heating rates obtained from wind-tunnel and theoretical techniques are adequate when boundary-layer blockage effects are considered, and that interpretation of flight measurements is subject to various phenomena that can have large effects on the data.

### INTRODUCTION

The design of the entry thermal protection system for a spacecraft is contingent upon the ability of the engineer to predict the heating environment and flow field that the vehicle will undergo as it traverses the atmosphere. Prior to the Apollo Program, the aerothermodynamic prediction techniques were based on theory and correlations of data that had been obtained on scale models tested in ground facilities or on small-scale payloads on flight vehicles. Some limited data had been obtained on the full-scale Mercury and Gemini spacecraft at velocities up to orbital speeds. The Apollo Program provided the first opportunity to make measurements on a full-scale vehicle at superorbital velocities.

The Apollo entry spacecraft (command module) was designed as an axisymmetric vehicle with an offset center of gravity to provide lift during entry. The vehicle entered the atmosphere of the earth at an angle of attack that presented an asymmetric shape to the airflow. As a result, the surrounding flow field was too complex to describe

analytically. An experimental investigation was required in order to define the aerothermodynamic environment for use in the design of the thermal protection system, an ablative heat shield. Wind-tunnel tests were conducted in various facilities to measure local pressures and heating rates on scale models of the Apollo command module at angles of attack ranging from  $0^\circ$  to  $35^\circ$ . The test conditions covered free-stream Mach numbers of 6 to 20 and free-stream Reynolds numbers (based on a body diameter) of  $0.03 \times 10^6$  to  $6.8 \times 10^6$ . The wind-tunnel data were valuable in the definition of the pressure and heating rates on the vehicle at specified angles of attack. Because the thermal-protection-system requirements were determined with analyses and wind-tunnel data, a flight-test program was used to verify the prediction techniques.

Four Apollo heat-shield-qualification flight tests were conducted (two at orbital and two at superorbital entry velocities), during which measurements of local pressure and heating rates were made. Comparisons of flight data with theoretical predictions and with wind-tunnel results were used to verify the design analysis. This report is a summary of the aerothermodynamic environment of the Apollo command module during entry into the atmosphere of the earth, and the following are described.

1. The Apollo spacecraft and heat-shield-qualification program
2. The verification of basic engineering aerothermodynamic predictions gained from the flight data
3. The aerothermodynamic knowledge gained from experience at superorbital entry conditions
4. The observations that offer a challenge to the analyst and the designer

## SYMBOLS

P	pressure, psia
$\dot{q}$	heating rate, Btu/ft <sup>2</sup> -sec
R	command module maximum radius measured from center-line axis, 6.417 ft
S	surface distance measured from center of blunt entry face, ft
$\alpha$	angle of attack, deg

### Subscripts:

cw	cold wall
meas	measured
pred	predicted
wb	with blowing

# CONFIGURATION AND FLIGHT TEST

## Entry Vehicle

The Apollo command module enters the atmosphere of the earth at a planned angle of attack of approximately  $20^\circ$  relative to the geometric center line of the vehicle. At this attitude, the spacecraft is a nonsymmetrical configuration with unique flow characteristics around the vehicle. The heat shield, or thermal protection system, is an ablator that covers the entire command module. An illustration of the entry vehicle (fig. 1) depicts the various flow phenomena associated with the following regions.

1. The blunt entry face, which is a segment of a sphere, contains a stagnation region, a high-heating area near the windward corner, and subsonic and supersonic flow regions.

2. The windward portion of the conical section is in an attached-flow regime.

3. The leeward portion of the conical section is in a separated-flow regime and undergoes low heating.

A toroidal fairing connects the blunt entry face and the conical section. This toroidal structure undergoes severe pressure gradients and a rapidly accelerating flow.

The heat-shield-qualification flight tests were conducted with a Block I design configuration (earth-orbital-entry Apollo command module), a predecessor of the lunar-mission spacecraft. A photograph of one of the recovered Block I flight test vehicles is shown in figure 2. The Block II, or lunar-mission spacecraft, is aerothermodynamically similar to the Block I configuration, except for such minor changes as truncating the conical section apex and removing the scimitar antennas and the leeward umbilical housing.

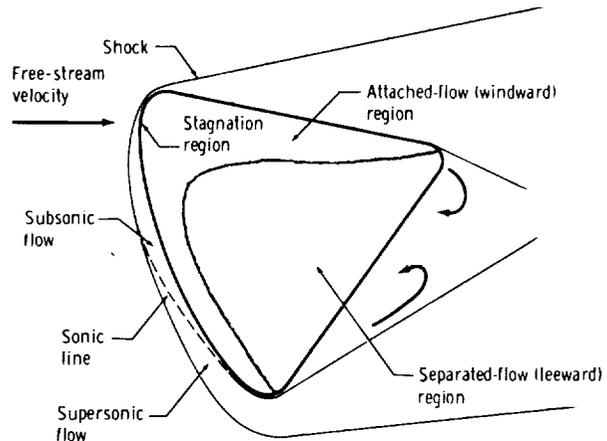


Figure 1. - Flow characteristics around the Apollo command module.

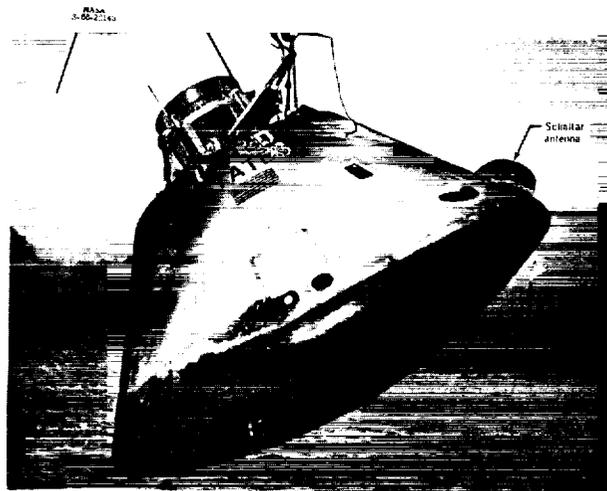


Figure 2. - A recovered Apollo test vehicle.

## Instrumentation

Flight measurements were needed in order to verify the aerothermodynamic environment predicted from wind-tunnel data and theories. The planning of the heat-shield-qualification flight tests included the strategic location of and the specifications for instrumentation around the test vehicle. Flight measurements were made by the use of pressure transducers, radiometers, and surface-mounted calorimeters located in the pitch plane and at various rays around the command module (fig. 3).

Only those spacecraft that were used for lunar-return-velocity tests had radiometers. The radiometers were located at the stagnation point and midleeward side of the blunt entry face, and two additional sensors were located on the conical section (one on the windward and one on the leeward side). The general features of the sensors are shown in figure 4.

Pressure transducers, located under the substructure, measured local pressures through a small hole in the ablator. Each radiometer, used to measure radiative heating rates, consisted of a thermopile behind a quartz window located at the bottom of a stepped hole in the ablator. Two types of calorimeters were used to measure total heat-transfer rates.

Asymptotic calorimeters, designed to measure heating rates less than  $50 \text{ Btu/ft}^2\text{-sec}$ , were located on the toroidal and conical sections. High-range slug calorimeters, located on the blunt entry face, consisted of a series of graphite wafers that were stacked to allow individual removal as the surrounding ablator receded. They were designed for use on the Apollo spacecraft. Unfortunately, the characteristics of these calorimeters required the use of extensive ground-test data in the calculation of heating rates from the wafer-temperature measurements. The slug-calorimeter design should be refined if measurements are required in ablative materials in the future.

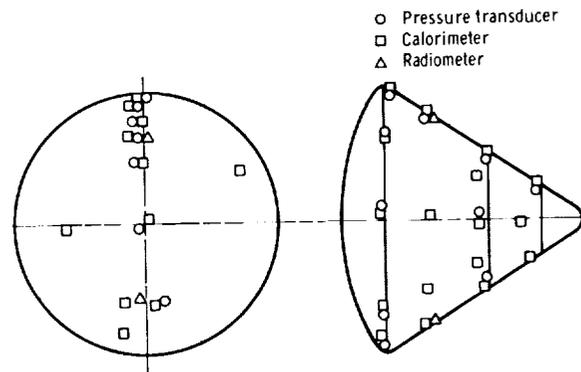


Figure 3. - The Apollo command module instrumentation locations.

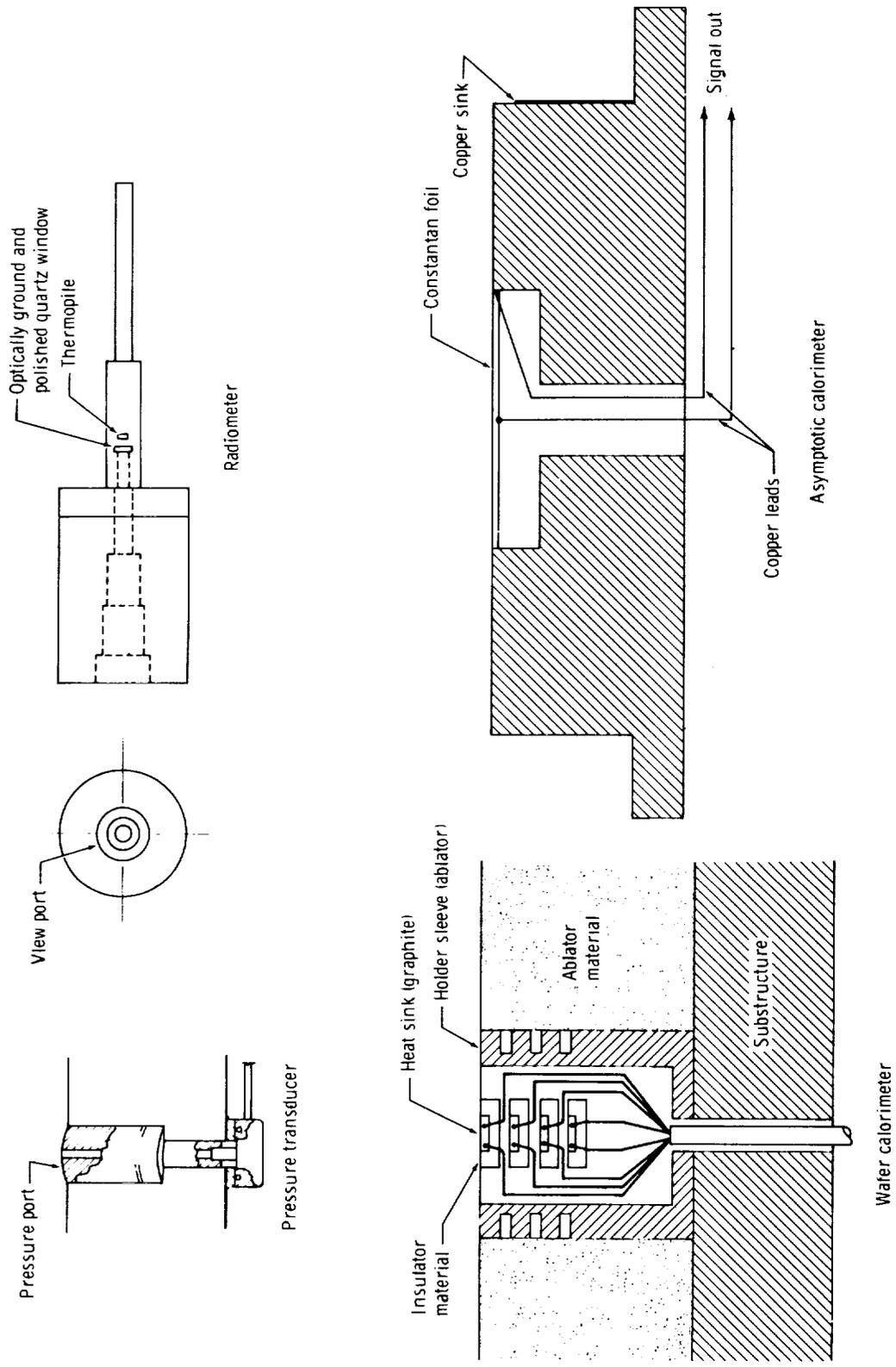


Figure 4. - A schematic of the aerothermodynamic instruments.

## Entry Trajectories

The entry envelope of the four heat-shield-qualification flight tests shown in figure 5 covered an initial entry-velocity variation of 10 000 ft/sec. The orbital-velocity qualification tests, conducted with spacecraft 009 and 011, entered at extreme ranges of entry angles and load factors that resulted in maximum g-levels of 14.3 and 2.4 ft/sec<sup>2</sup>, respectively. The corresponding maximum reference heating rates differed by a factor of 2, and the heat loads differed by a factor of 3, as listed in table I. The lunar-return-velocity qualification tests were conducted with spacecraft 017 and 020. Although spacecraft 020 entered at a velocity 3700 ft/sec less than was planned, the test provided valuable heat-shield-performance data. Three flights were performed in a low Reynolds number regime during the significant heating portion of the trajectories. However, spacecraft 009 underwent Reynolds numbers three times that of the other spacecraft because of its steep entry trajectory.

TABLE I. - HEAT-SHIELD-QUALIFICATION FLIGHT-TEST  
PERFORMANCE DATA

Spacecraft number	Angle of attack (nominal), deg	Inertial-entry flight-path angle, deg	Gravitational constant (max.), ft/sec <sup>2</sup>	Theoretical heating rate (max.), Btu/ft <sup>2</sup> -sec	Theoretical heat load, Btu/ft <sup>2</sup>
009	20	-8.58	14.3	164	6 700
011	18	-3.53	2.4	80	20 680
020	25	-5.90	4.6	209	28 274
017	25	-6.92	7.3	430	38 150

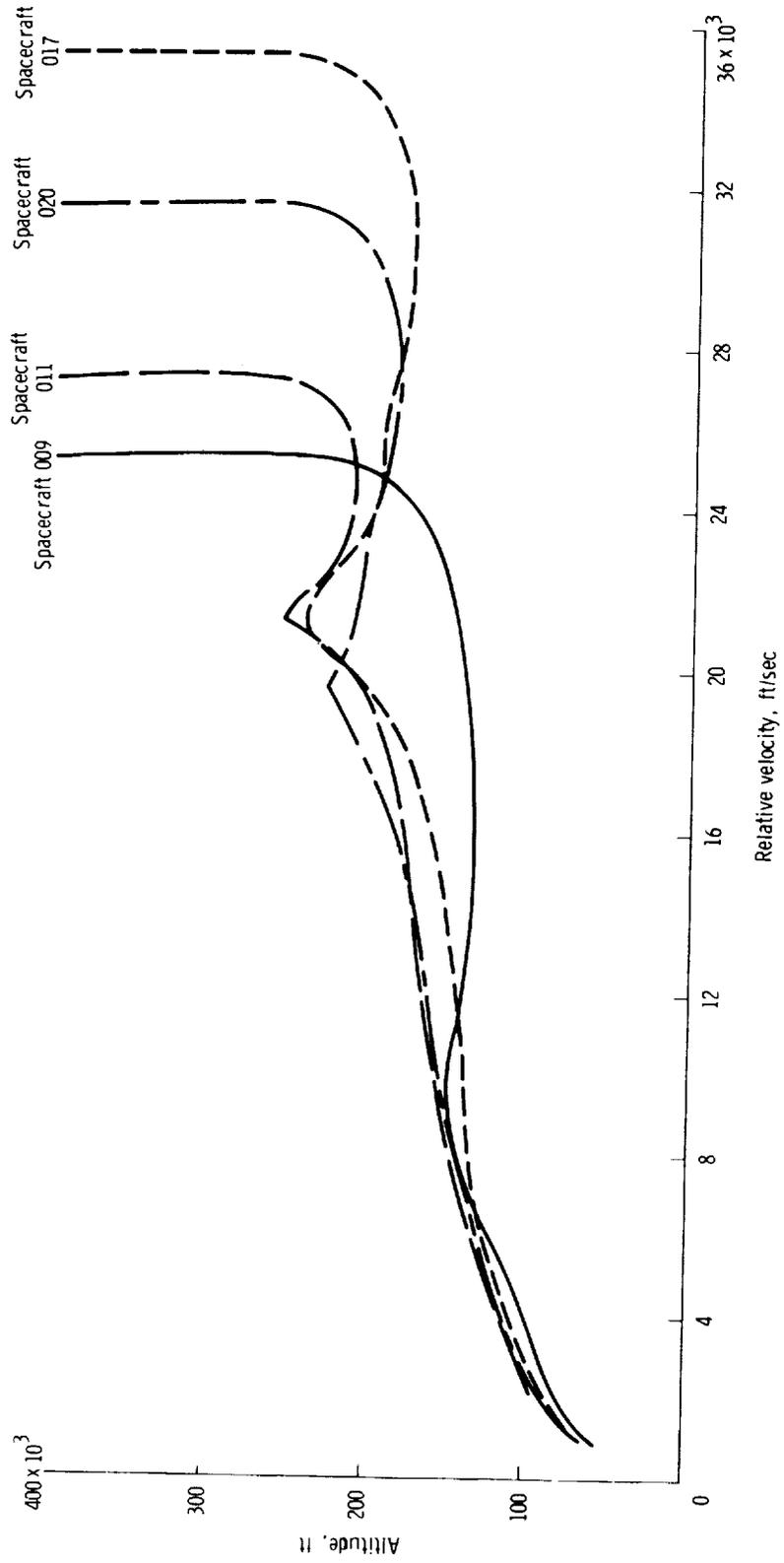


Figure 5.- The entry envelope of the Apollo heat-shield-qualification flight tests.

## RESULTS AND DISCUSSION

Examination of the flight data indicated that the measurements verified the use of preflight theoretical analyses and the use of ground-test data in the design of spacecraft for environments at least as severe as the Apollo entry environment. Extensive analyses of the data measured during orbital-velocity entries are given in reference 1, and data from superorbital-velocity tests are given in reference 2.

### Pressure

Blunt entry face. - In the absence of a three-dimensional flow-field solution for the Apollo command module at specified angles of attack, a wind-tunnel program was conducted to measure pressures around the entry configuration. Local-pressure values obtained from wind-tunnel tests are compared with the modified Newtonian theory (fig. 6). Both curves are similar in magnitude and in shape. However, the curves

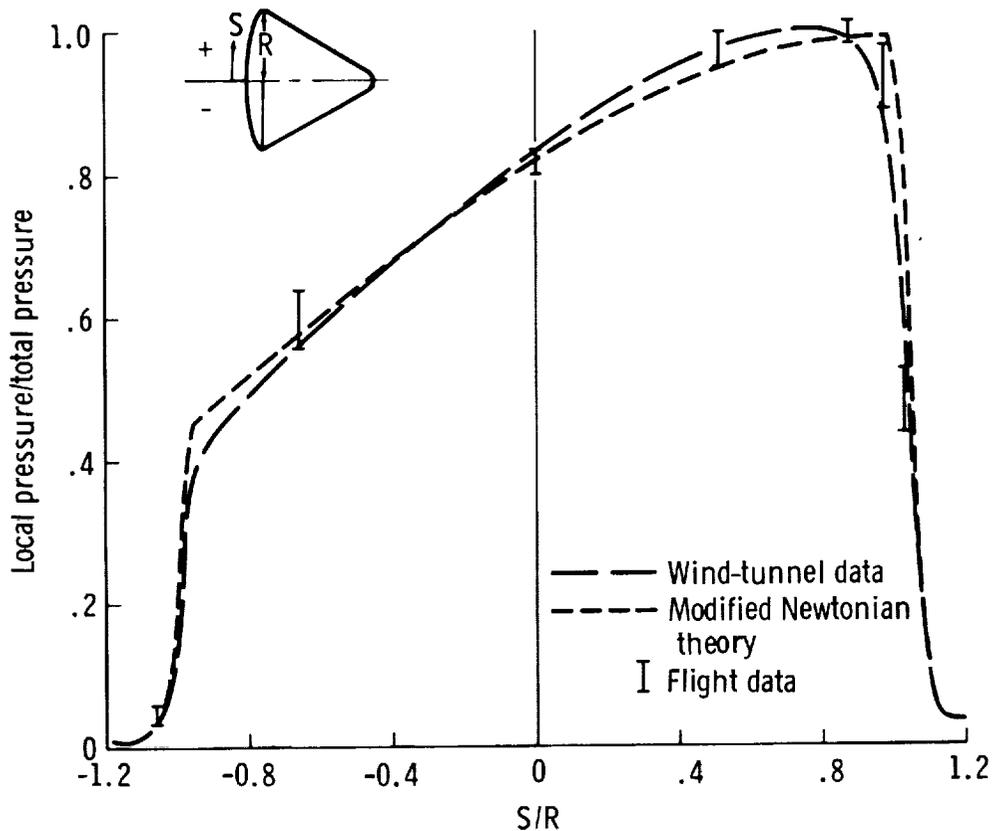


Figure 6. - A comparison of superorbital flight pressure measurements with wind-tunnel data and modified Newtonian theory for  $\alpha = 250$ .

differ in the distribution around the toroidal corners. Calculation of the local Reynolds number for the windward and leeward corners with the use of both the wind-tunnel pressure data and the Newtonian theory resulted in only a 10-percent difference in Reynolds numbers, even though the pressures differed by as much as 30 percent on the windward toroid. The flight data shown in figure 6 were obtained during the superorbital velocity entries with the spacecraft at a  $25^\circ$  angle of attack. The flight data, shown as bands that represent a large Mach number variation for a range of entry times, substantiate the previous distributions and lie between the two prediction methods on the windward corner.

Conical section. - The pressure measurements on the conical section generally agreed with the wind-tunnel predictions for the orbital flight of spacecraft 009 (ref. 1). However, during the entries of spacecraft 017 and 020, the conical pressure measurements were low (approximately one-third and one-half the predictions) during maximum heating. Some representative pressure histories measured on the windward conical section of spacecraft 020 are shown in figure 7. The differences between the flight pressure and the pressures based on wind-tunnel measurements that existed at the times of maximum heating might be caused by mass injection into the boundary layer from the aft-compartment ablator.

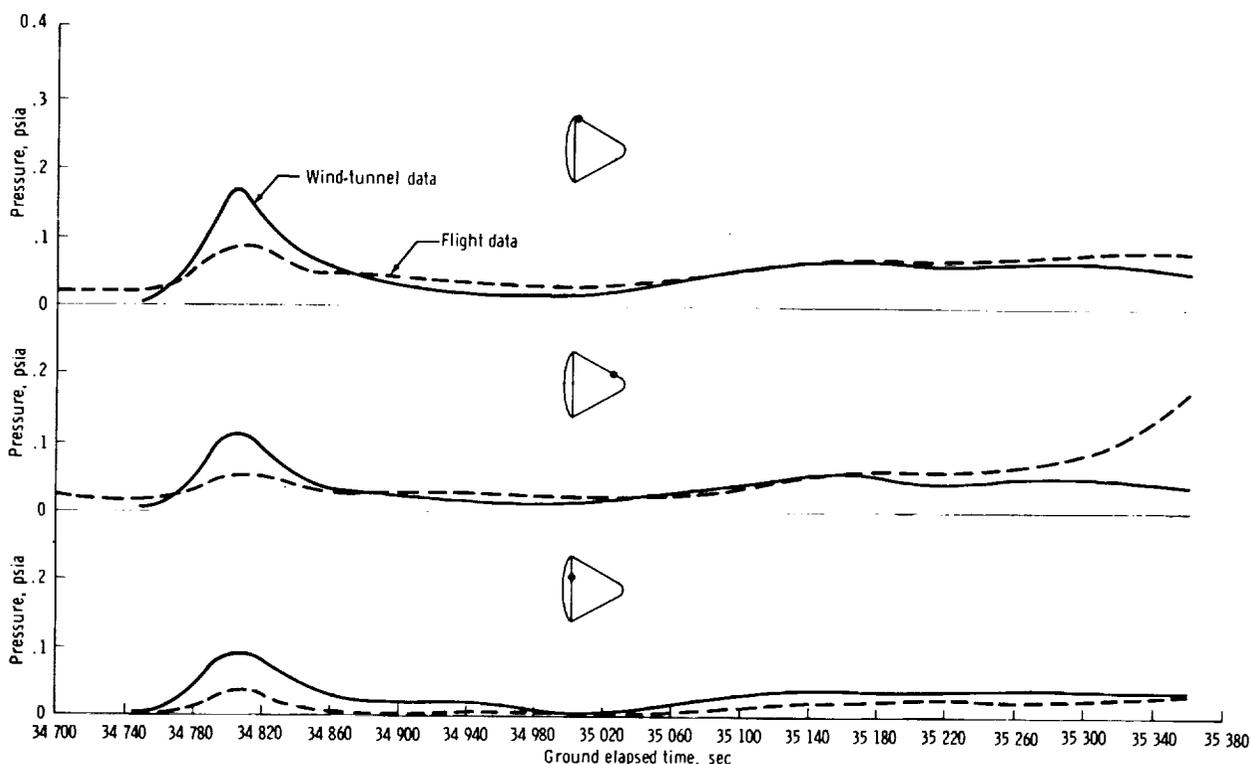


Figure 7. - Pressure histories on the windward conical section of spacecraft 020.

## Heating Rates

During atmospheric entry, the Apollo command module undergoes radiative and convective heat fluxes from the high-temperature air between the shock wave and the vehicle. Entries at orbital velocities result in negligible radiative heating rates. For the Apollo superorbital flight regime, radiative heating is approximately one-third of the total heating rate. The radiative heating is not affected significantly by outgassing from the ablator. However, convective heat transfer is lowered significantly by ablation products injected into the boundary layer.

Radiative heating. - The magnitude of the radiative heating to a blunt entry vehicle is a function of velocity, altitude (density), and the shock-standoff distance. For the analysis performed for Apollo entries, the stagnation-point-standoff distance was assumed to vary as it would for a sphere. This distance determines the thickness of air in the shock layer that radiates at both nonequilibrium and equilibrium conditions. The nonequilibrium radiation, originally considered to be a major source of heating, was found to be of minor importance for the Apollo entries when investigated by the use of shock-tube measurements. Knowledge of collision limiting, self-absorption, and interactions between radiation and gas flow has improved prediction techniques and accuracy during the course of the Apollo design and development.

The radiative-heat-transfer rate measured at the stagnation point of spacecraft 017 is given in figure 8. The flight data are in agreement with the theory of reference 3 calculated for visible and infrared radiation. The calculations include radiant energy losses through the use of a nonadiabatic system, and these losses are significant in the Apollo regime. At the time of maximum heating, the calculated peak nonadiabatic radiative heating was 76 percent of the adiabatic value.

There was no discernible response from the radiometers on the conical section during the heating portion of the entry trajectory. This lack of response confirms the prediction of negligible radiation to the conical region from air or from ablator particles. The radiometers did respond during the descent of the spacecraft on the parachutes while the excess fuel was being dumped. This response demonstrated the operational capability of the sensors.

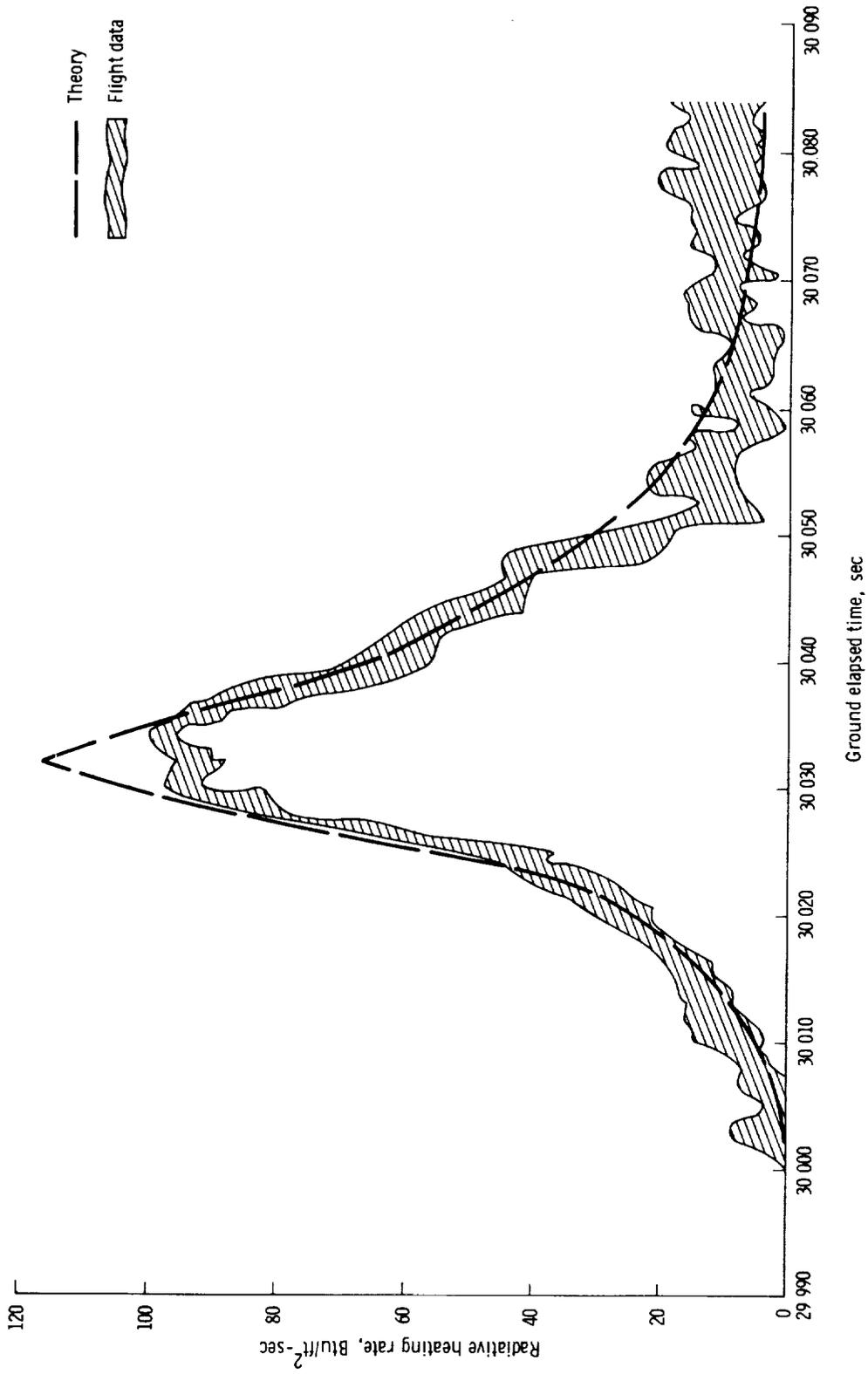


Figure 8. - The radiative heating rate measured at the stagnation point on spacecraft 017.

**Convective heating.** - The Apollo heat-shield-qualification test vehicles were the first full-scale spacecraft to be instrumented with calorimeters on the entry face. Heating rates to the blunt entry face were determined from the wafer-temperature flight measurements by the use of an empirical fit to postflight ground-test data. In the ground tests, the inflight wafer temperatures were used to control the heat inputs to the laboratory-monitored wafers. The resultant flight-derived heating rates that were measured at the stagnation point of spacecraft 017 are compared in figure 9 with the theoretical radiative plus convective heating rates adjusted for mass injection into the boundary layer (blowing). The comparison shows remarkable agreement. The wafer temperature became erratic at approximately 2000° F, resulting in an early termination of the heating-rate data. However, the 40 seconds of data collection were sufficient to illustrate the significant reduction in cold-wall heating because of ablation products injected into the boundary layer. The theoretical cold-wall heating rates were determined by use of the stagnation-point theory of reference 4 and the wind-tunnel measurements of heating rates at various locations around the command module; these rates are nondimensionalized because of the measured zero angle-of-attack stagnation-point value. The stagnation-point theory of reference 4 is based on the Apollo command module radius, 6.417 feet (one-half the maximum diameter), and is adjusted for the spherical-segment shape of the spacecraft. The cold-wall rates were used as an input to a charring and ablation computer program, designated STAB II (ref. 5), in order to calculate the heating rates adjusted for blowing.

The leeward conical section of the Apollo vehicle is in a separated-flow regime wherein the heating rates are low and the ablation material does not char. The flight data obtained from asymptotic calorimeters agreed with the predictions based on 2 percent of the calculated stagnation-point value. In some locations, measured rates were as low as 1 percent. No effects of protuberances were observed, and only momentary responses to reaction-control-engine firings were detected.

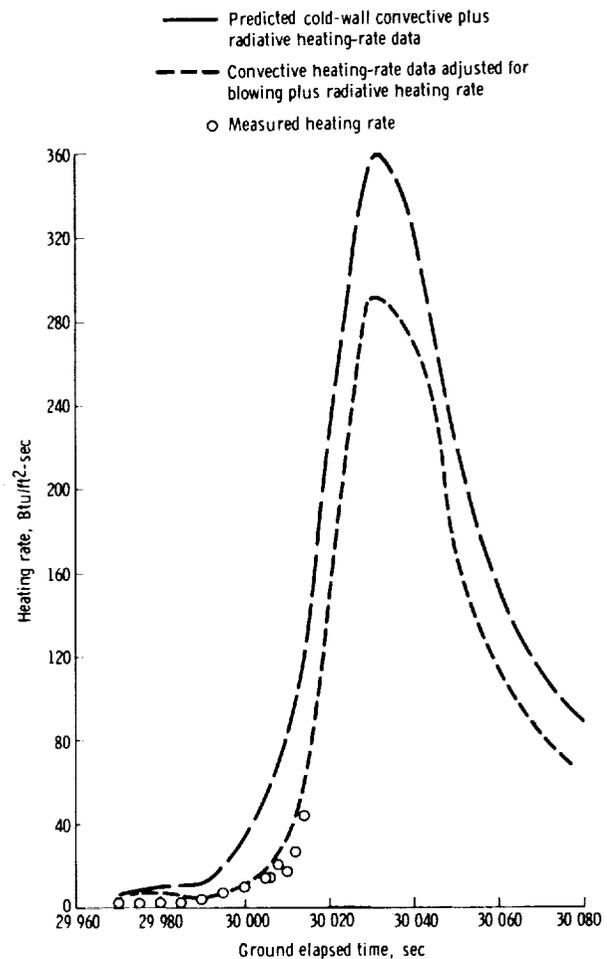


Figure 9. - The stagnation-point heating-rate history determined from wafer calorimeter measurements on spacecraft 017.

The windward conical section is in an attached-flow regime wherein the heating is sufficiently high to char the ablator. The cold-wall theoretical heating rates, adjusted for blowing, agreed with the flight measurements obtained during entry from orbital flights (ref. 1). However, the adjusted cold-wall predictions exceeded the superorbital-flight measurements (ref. 2). This discrepancy corresponded to similar behavior observed in the pressure measurements taken in this region.

## OBSERVATIONS

### Windward Conical Section

The local pressures and heating rates measured on the windward conical section were below the predicted values for the superorbital entries. The flight data obtained during orbital entries, however, were in agreement with the predictions based on wind-tunnel results and theoretical cold-wall heating rates. When all of the qualification-flight data were compared, the orbital-entry heating rates of spacecraft 009 actually were higher than the rates measured on spacecraft 017, despite the 10 000-ft/sec lower velocity of spacecraft 009. This phenomenon prompted an examination of the ablative material on recovered spacecraft 009 and 017. Examination of core samples taken at various locations around the vehicles revealed that the char penetration was three times deeper on the aft compartment of spacecraft 017 than on that of spacecraft 009. This observation enhanced the suspicion that upstream blowing might disturb the flow over the conical section. Calculations performed by the use of the charring and ablation computer program revealed a correspondence between the times of high mass-injection rates at the stagnation point and the low pressures measured on the windward conical section.

In an effort to predict the low conical-section heating rates during superorbital entries, the cold-wall heating rates were adjusted by (measured pressure/predicted pressure)<sup>1/2</sup>. This adjustment lowered the cold-wall heating-rate predictions by 60 to 70 percent and brought the heating rate with blowing  $\dot{q}_{wb}$  into agreement with the measured rates, as shown in figure 10. However, the capability to predict the lower pressure does not exist currently. Not only is a three-dimensional flow-field solution required for description of the environment around the vehicle, but the solution is complicated by the ablative process.

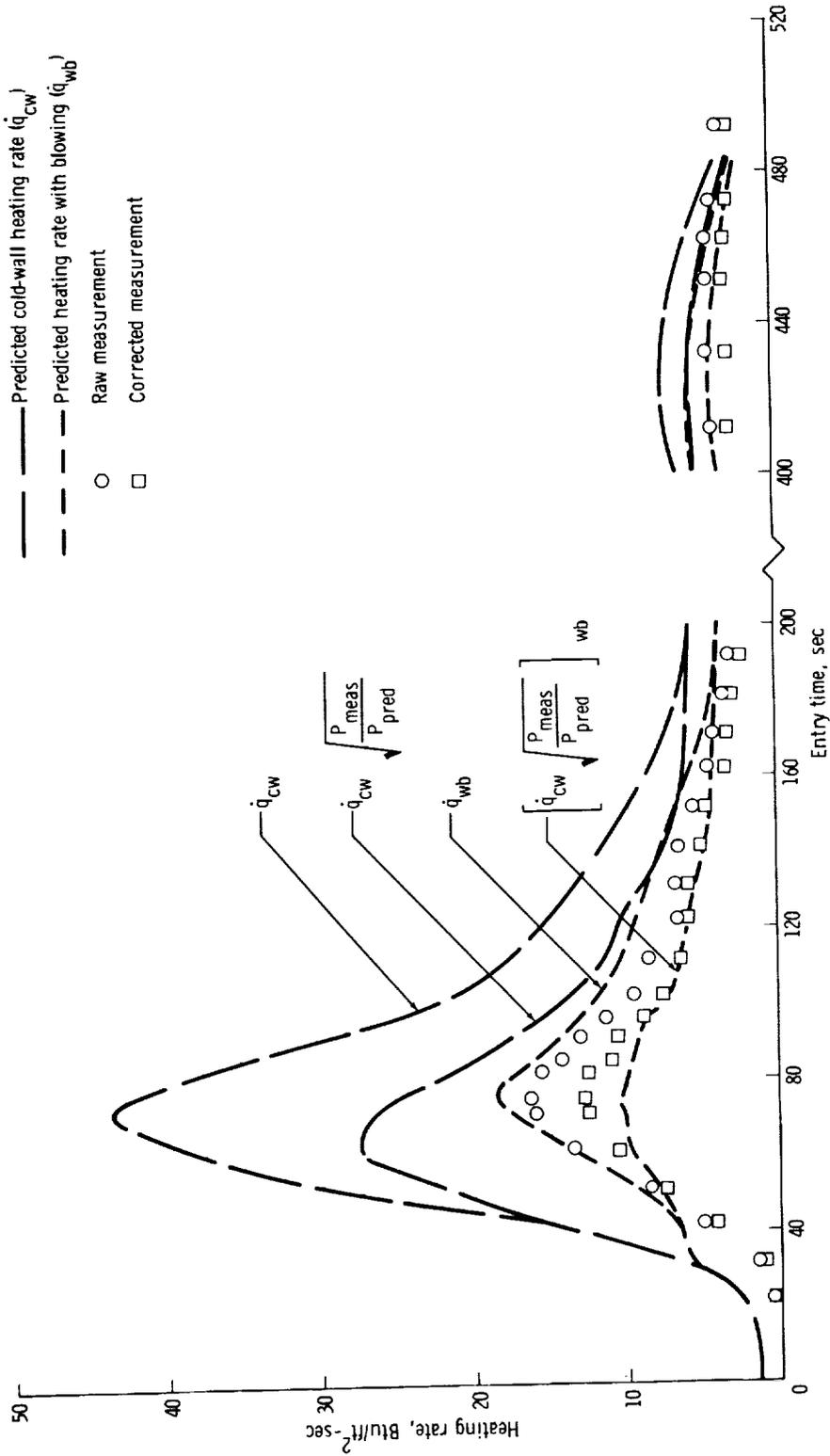


Figure 10. - A typical heating-rate history measured on the windward conical section of the Apollo command module during superorbital entry.

## Calorimeter Interpretation

The asymptotic calorimeter is designed to maintain a body temperature of 400° F or lower in order to have a meaningful ground-test calibration. This body temperature necessitates consideration of the temperature difference between the sensor and the surrounding hot-heat-shield material. The method of Woodruff, Hearne, and Keliher (ref. 6) was used to calculate the nonisothermal effect. Because the nonablating sensor was located in an ablative material, the effect of discontinuous mass injection was investigated (based on the method of Hearne, Chin, and Woodruff (ref. 7)). These effects were found to be three times as large near the toroid section (6 to 12 percent) as near the apex (2 to 4 percent).

## CONCLUSIONS

The experience gained from the Apollo Program resulted in greater confidence in aerothermodynamic prediction techniques and extended future prediction capability to include higher velocities. Analysis of the data obtained on the heat-shield qualification flight tests provided the following conclusions, which are applicable to future spacecraft design.

1. Wind-tunnel data and modified Newtonian theory can be used to predict local pressures on a blunt entry face during hypersonic flight.
2. Radiative heating rates on the blunt entry face can be predicted by the use of theoretical values for visible and infrared radiation at velocities as great as those involved in superorbital entries.
3. Air and ablation-products radiation to the conical section are negligible.
4. Convective heating rates on the blunt entry face can be predicted by the use of wind-tunnel results and the stagnation-point theory of Detra, Kemp, and Riddell when adjusted for blowing.
5. Heating rates measured in separated-flow regions agree with wind-tunnel-based predictions (2 percent of the stagnation-point theory calculated for the Apollo spherical-segment entry face).
6. The pressures measured on the windward conical section agree with wind-tunnel results for orbital entries, but are below wind-tunnel-based predictions for superorbital flights. The low pressure is attributed to upstream blowing. A technique to predict the low pressure has not been developed.
7. Convective heating rates in the attached-flow regions agree with predictions lowered by  $(\text{measured pressure}/\text{predicted pressure})^{1/2}$  and then adjusted for blowing.

8. Calorimeter measurements obtained in an ablative environment require corrections of nonisothermal effects and discontinuous mass injection.

Manned Spacecraft Center  
National Aeronautics and Space Administration  
Houston, Texas, June 7, 1971  
914-11-20-10-72

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